NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Technical Report 32-955

Mariner Mars 1964 Temperature Control Hardware Design and Development

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Contents

Contents (contd)

Figures

Contents (contd)

Figures (contd)

Abstract

The temperature control system on the *Mariner Mars 1964* spacecraft, including both passive and active controls, is described. Passive controls comprised (1) special surfaces and finishes and (2) thermal shields; active controls were thermostatically actuated louvers. The evolution of the design is traced from inception to final configuration, including design criteria and fabrication techniques. The criteria for selecting surface finishes and materials are discussed in relation to spacecraft requirements and space-simulator testing conditions. Recommendations are made that may be applicable to the design of temperature control hardware for future projects.

Mariner Mars 1964 Temperature Control Hardware Design and Development

I. Introduction

The objectives of the *Mariner Mars 1964* flyby mission of the planet Mars were to obtain scientific information on interplanetary space and on conditions near Mars, television pictures of the Martian surface, and occultation data from spacecraft radio signals as they were affected by the atmosphere of the planet. The *Mariner IV* spacecraft was launched on November 28, 1964, and encountered the planet on July 14, 1965, successfully completing the mission objectives. The spacecraft was still operating nominally when the mission was terminated on October 1, 1965, at which time the spacecraft radio transmission was switched to the low-gain antenna, permitting it to be tracked from Earth until mid-1967.

The *Mariner Mars 1964* spacecraft, which included 138,000 parts and weighed 575 lb, was 9½ ft high and spanned 22½ ft with solar panels deployed. The basic octagon structure was approximately 4 ft wide by 1½ ft high and contained seven bays of electronics and the midcourse motor. The spacecraft power was provided by four solar panels, hinged 90 deg apart from the octagon with a total area of 70 square ft and approximately 28,000 photovoltaic solar cells. Figures 1 and 2 locate seven scientific experiments, communication antennas, and other salient features.

The temperature control system comprised 40 separate assemblies covering nearly 95% of the spacecraft's basic structure. The complete system weighed 18 lb and contained approximately 1200 parts. A chronological description of the development and fabrication is presented in this report.

II. Temperature Control System

The purpose of the temperature control system was to maintain temperatures of subassemblies, assemblies, and components of the spacecraft within specified bounds. Passive control, consisting of special finishes and coatings and thermal blanket shields, was used where it could adequately meet the requirements, and active control, consisting of thermostatically actuated louvers, was added where necessary.

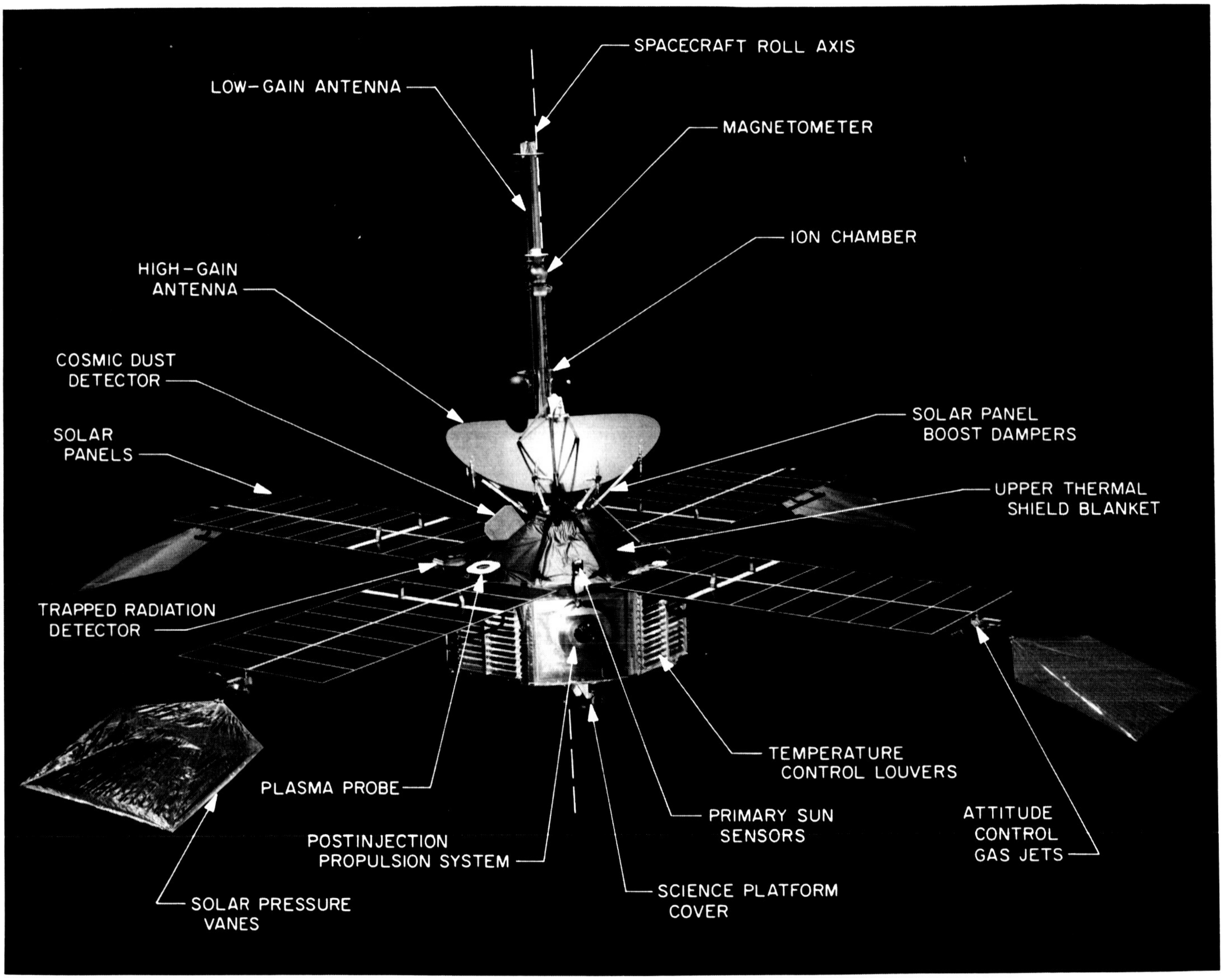

Fig. 1. Spacecraft configuration (top)

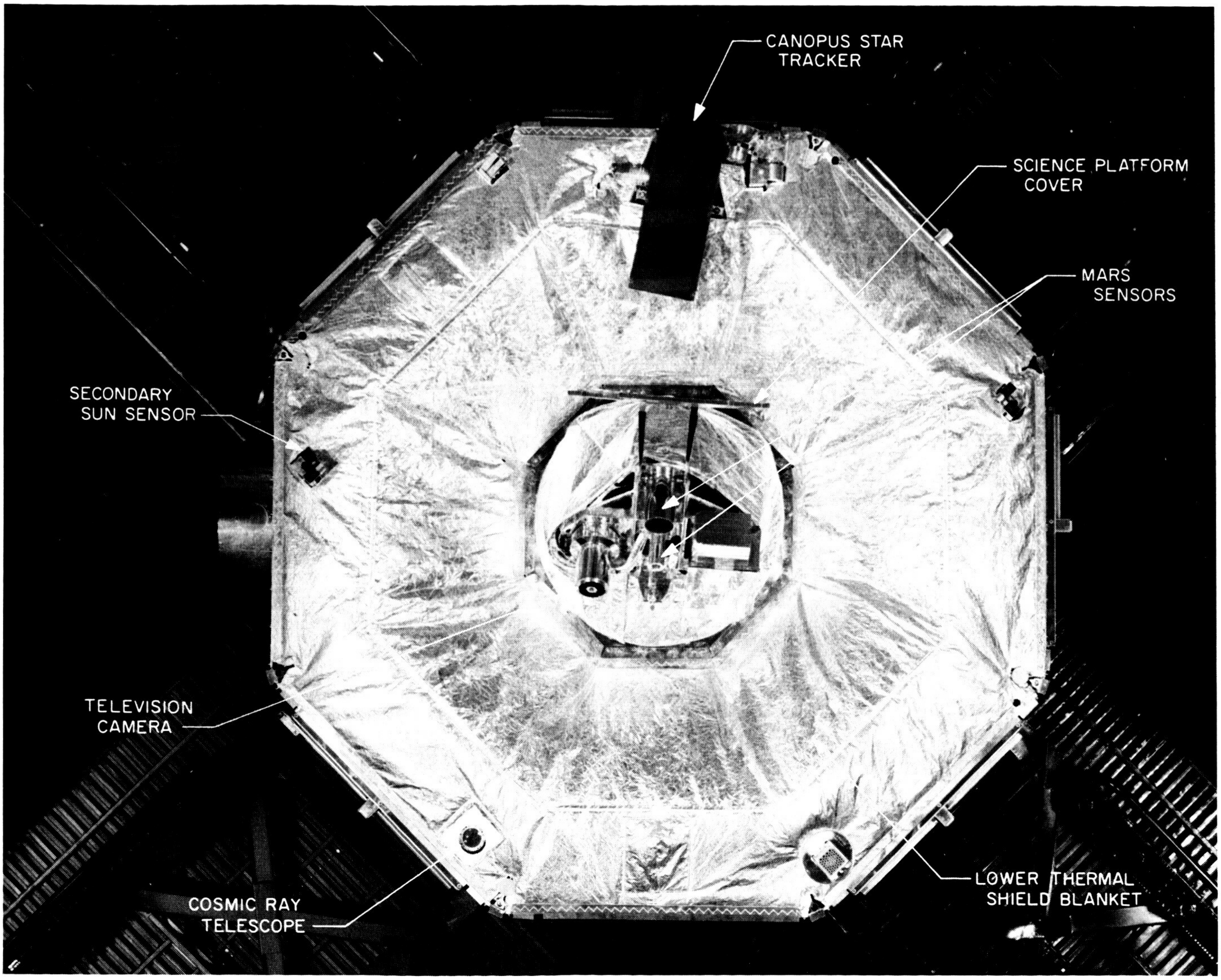

Fig. 2. Spacecraft configuration (bottom)

Important features of the *Mariner Mars 1964* temperature control design were (1) isolation from solar heating, (2) minimum internal resistance to radiative and conductive heat transfer, and (3) variable emittance, provided by the louvers, on most electronic bays. Protection from the variable solar input was provided by a multilayer aluminized Mylar thermal shield on the sunlit portion of the spacecraft. Localized internal hot spots were prevented by using good thermal conduction joints and by treating the interior surfaces of the spacecraft with high-emittance coatings.

The louvers were provided for active temperature control as a result of the experience with *Mariner II*.[1] By varying the effective emittance of the louvered areas, the *Mariner IV* louvers suppressed the temperature excursions caused by changes in the solar intensity, by thermal shield leaks, and by radiation from the solar panels to the electronic assembly faces.

The unlouvered portions of the sides of the spacecraft were covered with low-emittance thermal shields to minimize heat losses. A multilayer aluminized Mylar thermal shield covered the lower (shaded) side of the spacecraft to minimize heat losses. A second lower thermal shield on the scan science instruments slaved their temperatures to that of the basic octagon structure.

External science instruments and appendages were separated from the basic spacecraft structure and were passively controlled by the selection of appropriate materials and, where appropriate, by shading from the Sun.

An optimum balance of passive and active control was not established until late in the design phase. As a result, a universal temperature-control louver assembly, which could be mounted on any electronic chassis, was established. The final decision as to which electronic chassis required active control was made late in the program, with only minor changes required to incorporate the revision.

III. Surface Finishes and Materials

A. Surface Selection Criteria

The *Mariner Mars 1964* thermal design and temperature control surface selection was basically conservative. Margins for uncertainty in surface property measurements and behavior in space were allowed, both in design and in surface selection. Exotic, unproven, unrepairable, and easily damaged materials were not considered for use. For example, vacuum-deposited aluminum has properties somewhat superior to those of polished aluminum surfaces. However, the cost of repair in money, time, and component reliability in the event of damage far outweighs such property advantages. The validity of this conservative philosophy in surface selection was verified when a space simulator incident deposited substantial amounts of oil on the thermal surfaces of one of the spacecraft. Nearly all of the surfaces involved could be, and were, adequately cleaned with assurance of reliable mission performance.

The criteria for selecting temperature control coatings included the proper combination of radiative properties and reliable behavior in the space environment, as well as satisfactory performance in the terrestrial environment. (The latter criterion includes durability and cleanability for handling, behavior in a simulator, and behavior in space as affected by prelaunch conditions.) While severe terrestrial conditions were not considered (i.e., long outdoor exposure or salt spray), the problems connected with fabrication, handling, and assembly were real constraints. For example, bare aluminum can generally be used without surface protection, while bare silver, although satisfactory in the space environment, would tarnish in the terrestrial environment. Because exterior surfaces were painted early in the assembly sequence, handling could damage or contaminate the temperature control surface and cause poor performance in space.

Also considered in the design was the possibility of a different response to the solar spectrum and to the spectrum of the solar simulator in the 25-ft space chamber at the Jet Propulsion Laboratory (JPL), in which the temperature control system was tested. The possibility of such a simulator–Sun spectrum mismatch was not a general constraint, however.

B. Surface Materials

Surface materials used included bare or plated metals; conversion coatings; and paints.

*1. **Metals.*** Bare metals such as aluminum, silver, gold, and rhodium have low solar absorptance and low emittance and behave as thermal isolators, minimizing both the absorbed solar energy and the energy radiated from the vehicle to space. Metallic surfaces other than aluminum (i.e., magnesium, copper, stainless steel, etc.)

[1] The *Mariner II* spacecraft carried out a successful flyby of the planet Venus on December 14, 1962, after a flight of 109 days, although flight temperatures were considerably higher than the flight spacecraft simulator test data.

are normally gold-plated to provide either low emittance or corrosion protection or both. These surfaces can then be painted to increase the absorptance or emittance as required.

For the attitude control jets, a lower emittance was required than was obtainable from bare stainless steel. Rhodium plating was selected in preference to gold since it is thermally gray (i.e., the reflectance is nearly constant at all wavelengths in the solar region) and is thus less sensitive than gold to simulator–Sun spectrum mismatch.

2. Conversion coatings. A commercial dip-process conversion coating, Dow 7, was used on machined magnesium parts where a high emittance was desired or where emittance was unimportant to the thermal design. The process was controlled by matching the material being coated to a color reference sample known to produce a minimum emittance of approximately 0.70. In addition to this minimum emittance, the conversion coating provided a degree of corrosion protection.

A brush-on conversion coating, Dow 19, was used for repair and touch-up of remachined areas. Since the Dow 19 produces a very thin coating, emittance of approximately 0.2 resulted. Its use was limited to situations where, either by reason of the thermal design requirements or of the small area ratios involved, this low emittance was not a detriment. The Dow 19 low emittance, coupled with the degree of corrosion protection provided, was used to advantage on the areas adjacent to the spacecraft feet that mated with the *Agena* adapter. A low emittance was desirable to minimize heat loss from those areas.

Instrument design and calibration of the *Mariner Mars 1964* ion chamber required that temperature control be achieved without adding significantly to the particle absorption properties of the sphere. A conversion coating for the stainless steel ball conforming to specification MIL C-13924 Class II was used to produce a solar absorptance ≈ 0.90 and a surface emittance ≈ 0.70. It permitted satisfactory temperature control without adversely affecting the instrument performance. Because this was a passive conversion coating, the properties produced were sensitive to initial metal surface conditions. In order to provide reproducible, uniform properties as listed above, the surface was liquid-honed prior to application of this conversion coating. One of the flight ion chambers tested with the first flight spacecraft ran considerably hotter than had the previous instruments, indicating that the surface emittance was lower than the design value. The emittance was checked with the Lion Portable Emissometer and found to be low. Under close scrutiny, the coating appeared thin, possibly as a result of poor application or having been scuffed during handling. A patch of high-emittance paint was applied to the electronic chassis attached to this particular ion chamber sphere to compensate for the low sphere emittance.

3. Paints. Both black and white paints were used; others included metal-filled paints and one pigmented with green chromium oxide.

a. White paints. Exterior surfaces of the electronic chassis, under the louvers, required only a high emittance for cruise conditions. However, it was necessary to minimize the solar load input that could occur during pre-Sun-acquisition and midcourse maneuver operation when sunlight fell on these surfaces. Since the maximum exposure time was less than 10 hours, long-term ultraviolet stability was not a prime concern. The original thermal design included the use of an inorganic white paint on these chassis faces. This inorganic paint, primarily developed for ultraviolet stability, had a low solar absorptance and a high emittance. After the painting of the initial Temperature Control Model hardware, it became apparent that extreme precautions would be required in order to provide reliable adherence to these large surface areas. Re-evaluation of the thermal requirements indicated that an organic paint with slightly higher solar absorptance and somewhat lower emittance but with far superior adhesion and durability properties would perform satisfactorily. This paint, designated PV-100, was subjected to ultraviolet tests and found also to be suitable for some sunlit applications for the *Mariner Mars 1964*, including the high-gain antenna feed and the solar plasma experiment. The inorganic white was used only on the cosmic ray telescope radiator, where the extremely high emittance and low solar absorptance were critical to the instrument operation.

A gloss-white epoxy, which was more durable than the PV-100, was applied to the basic structure longerons because the spacecraft lifting fixtures were attached to these longerons and a probability of handling damage existed. The thermal properties of the epoxy were not used because the design resulted in polished aluminum shields over most of the coated area.

b. Black paints. The back side of the *Mariner Mars 1964* solar panels was originally painted with a satin-black polyurethane paint. This material had been successfully used on both the *Mariner II* and *Ranger* spacecraft. Subsequent problems with stray light reflections into the Canopus tracker required that this paint be changed to a more specular gloss black.

Most of the internal components of the spacecraft were coated with a flat black epoxy paint that had a high emittance and provided maximum radiation coupling inside the basic spacecraft structure.

c. Other paints. Metal-filled paints have a solar absorptance-to-emittance ratio approximately equivalent to that of black paints. However, since both the solar absorptance and emittance are substantially lower than that of black paint, thermal transients during an eclipse period are substantially reduced. Applications of these paints, which are sensitive to handling, were restricted to those areas that could be adequately protected from damage or readily repaired.

The *Mariner Mars 1964* high-gain antenna presented an interesting temperature control problem. It was necessary to maintain the maximum temperature below the limits of the adhesive used in the honeycomb fabrication and to provide optimum temperature for minimum thermal distortion at planet encounter. The surface coating also had to avoid concentrating reflected solar energy on the antenna feed if the Sun should become incident along the parabolic axis.

A black paint would have produced temperatures in excess of the structural limits of the antenna near Earth and a white one would have resulted in low temperatures with accompanying thermal distortion at Mars. Poor lateral thermal conduction of the thin-gauge skins eliminated the possibility of a practicable mosaic approach. Grays formulated with combinations of white and black pigments were difficult to control, in addition to being subject to ultraviolet darkening in space.

An unpublished paper by G. A. Zerlaut[2] reported that paints pigmented with green chromium oxide (Cr_2O_3) have solar absorptance ~ 0.70 and are stable to ultraviolet. A polyurethane paint with this pigment was formulated and found to be satisfactory. This paint was also used on the omnidirectional antenna ground plane and the ion chamber Sun shade.

[2]Now at the Illinois Institute of Technology Research Institute, Chicago, Illinois. The work on Cr_2O_3 was done at the Marshall Space Flight Center of the National Aeronautics and Space Administration.

d. Paint application. Temperature control paints were applied by conventional spraying and brushing techniques. Brush application was used only for small rework areas or when adequate masking was impractical or impossible owing to potential hazard to sensitive adjacent surfaces or components and when the added uncertainties in radiative properties resulting from brushing were insignificant to temperature control. Spraying was accomplished in a standard paint spray booth using conventional airbrush equipment.

Controlled access and good housekeeping practices were maintained in the painting, preparation, and drying areas, but "clean room" conditions were not considered necessary. Some components that had been previously cleaned and had sensitive areas sealed were coated without jeopardizing cleanliness.

The necessary removal of parts from handling fixtures and protective packaging for painting made proper personnel training and area access control imperative.

Minimum thickness required for optimum thermal properties was determined for each paint early in the program. Each part or a simultaneously prepared control sample was checked with a Permascope[3] (nondestructive eddy current device) to assure that the required minimum thickness had been achieved without having excess amounts applied. This information, along with operator experience, made it normally possible to maintain paint thickness tolerances to ± 0.001 in.

IV. Thermal Shields

A. Design Criteria

Shielding, another passive control, was designed to maintain temperatures within the correction capability of the active louver assemblies. Care was taken to design around the limitations of the solar simulator in JPL's 25-ft space chamber. Thermally "gray" coatings were used wherever possible on sunlit surfaces to render them insensitive to simulator–Sun spectrum differences. To minimize the effect of decreasing Sun intensity, the solar dependence of the bus had to be small and known. The Sun-oriented upper thermal shield was a barrier preventing solar radiation as a changing source of bus heat input. The peripheral and lower shields, primarily exposed to low-temperature space, minimized the spacecraft's

[3]Permascope Type EC2 T2.3, Twin City Testing Co., Tonawanda, N. Y.

heat loss to that sink. All normally shaded shielding was capable of withstanding short periods of solar radiation, such as occurred during midcourse maneuver. Although temperature-control shielding requirements were considered during preliminary design, detailed shields were not included in the final structural design of the spacecraft. As the temperature control shielding was defined, it was incorporated on the spacecraft with minimum effect on weight, structure, components, operations, and configuration envelope. Design of the thermal shields included consideration of fabrication and handling, spacecraft systems testing, component accessibility, and final assembly procedures. Structurally, the shields were designed by the launch and ground-handling criteria. By definition, the final design was a compromise between the theoretical and the practical.

B. Overall Design

To satisfy the thermal and mechanical requirements, a flexible, superinsulation-type multilayer blanket was used on the top and bottom of the spacecraft, rather than metal shielding, since the blankets weighed less per square foot and imposed fewer attachment restraints on the spacecraft.

The blankets were fabricated of wrinkled aluminized Mylar sheets at a density of approximately 140 layers per inch. The blanket thickness could be varied to achieve an optimum thermal barrier. Little structural support was necessary, because the blanket was lightweight and flexible and could conform in geometrically difficult areas. The shield configuration was defined by tailoring patterns directly from a spacecraft (Fig. 3, top; Fig. 4, bottom); the full-size paper patterns on the spacecraft allowed visual and physical verification of the design objectives. The paper patterns were transferred to hard cardboard and then to full-size engineering drawings.

The case shields, which conformed to areas that were geometrically more simple, were defined by conventional engineering drawing methods.

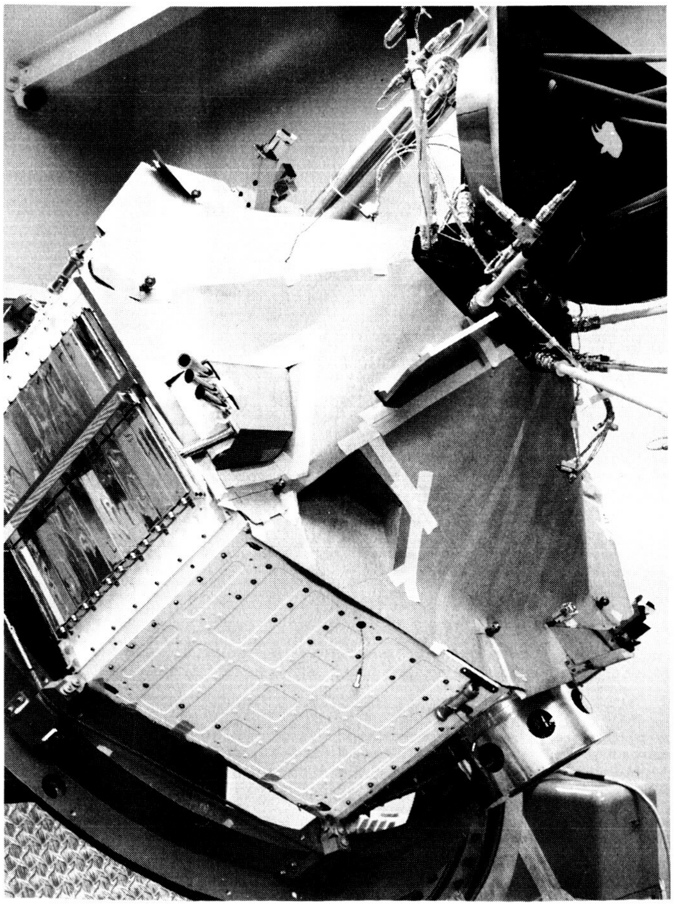

Fig. 3. Upper thermal shield pattern

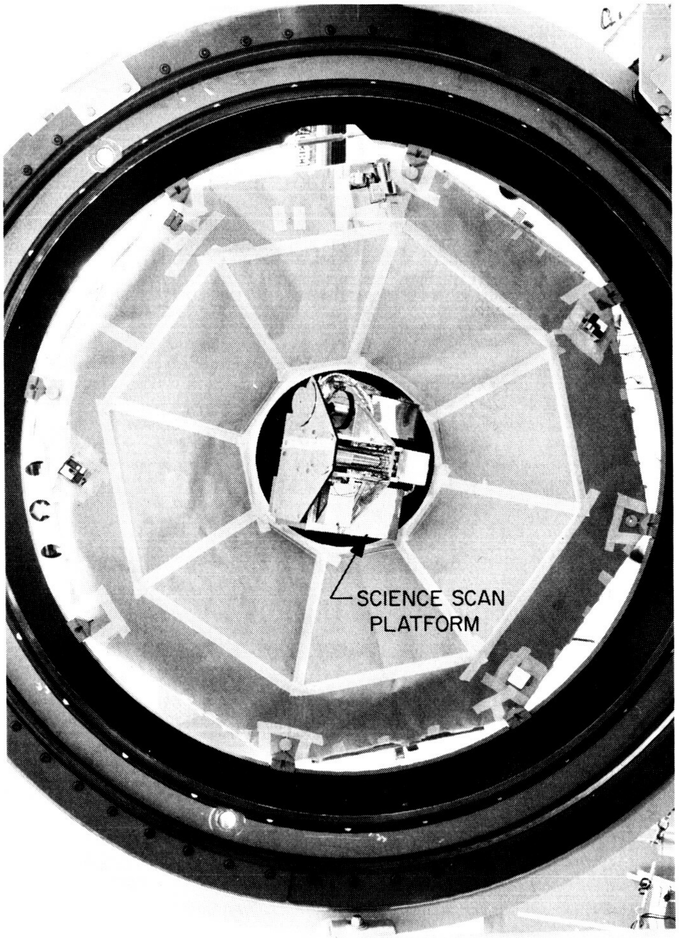

Fig. 4. Lower thermal shield pattern

C. Materials

The initial *Mariner Mars 1964* thermal design required that the outer layer of the upper thermal shield be insensitive to differences between solar and space-simulator energy distribution and that it not reflect appreciable heat to other portions of the spacecraft. Since the preliminary design of the spacecraft antenna support structure required that the upper thermal shield conform to a compound curvature, coated fabrics were not seriously considered. Instead, preformed black polymeric materials that could withstand the temperatures produced by an insulated black surface (approximately 260°F) were considered for this application. Preliminary evaluation was performed on a black-pigmented Teflon dispersion that could be sprayed on a shaped mandrel, fused to a film, and then stripped off. When the antenna support structure design was completed and the compound curvature eliminated, a tailored, fabric-covered shield became more practicable than one of preformed Teflon.

Samples of a dacron fabric coated with a black-pigmented silicone rubber were obtained and evaluated. A special lot of coated fabric sufficient to fulfill the needs of the *Mariner Mars 1964* program was then ordered and used. It was found that the coated fabric contained substantial amounts of volatile materials, and it was necessary to precondition the material before fabricating it into thermal shields. Subsequent chamber testing revealed that this preconditioning was not sufficient to completely outgas the material, and additional conditioning was performed on the fabricated shields.

While minimum emittance would be desirable on the outer layer of the lower thermal shield, the possibility of incident solar energy during the midcourse maneuver prevented the use of aluminized polymer films with the aluminum side out. If this condition occurred, it would result in temperatures beyond the limits of the blanket materials. It was therefore necessary to provide an outer layer that would have at least a moderate emittance. FEP Teflon, Type A, was known to be stable to ultraviolet radiation. Aluminized 5-mil film had been used as the outer layer of the sunlit thermal shield for *Mariner II*. For the *Mariner Mars 1964* application, 5-mil film had an undesirably high emittance (≈ 0.82). It was estimated that ½-mil aluminized Teflon would have an emittance ≈ 0.40 and would be suitable thermally. However, ½-mil film was extremely difficult to handle because of poor tear resistance. Therefore, 1-mil aluminized film with an emittance of approximately 0.60 was used as a suitable compromise of acceptable physical properties and moderate emittance.

Cabling outside the basic structure was wrapped with aluminized polymer film with the aluminum side out, to minimize absorbed solar energy and radiation to space. Originally it was planned to use aluminized Mylar, as had been done on previous missions. However, it was found during simulator testing that ultraviolet radiation penetrating discontinuities in the aluminum layer caused physical damage to the film. The Mylar was replaced with aluminum-coated, white-pigmented Tedlar, since the literature indicated that Tedlar was less subject to physical damage by ultraviolet radiation than Mylar. In addition, commercially available white Tedlar contains a pigment that is a strong absorber of ultraviolet, thus limiting the ultraviolet penetration and possible damage to a very shallow layer.

D. Detailed Design

1. Upper shield. The upper thermal shield blanket insulated the spacecraft octagon from the solar heat. The inner surface remained at a constant near the bus average (75°F at Earth mode) while the outer, Sun-exposed surface attained 260°F.

The blanket consisted of 30 layers of vacuum-metallized ¼-mil duPont Mylar polyester film, Type C. The Mylar

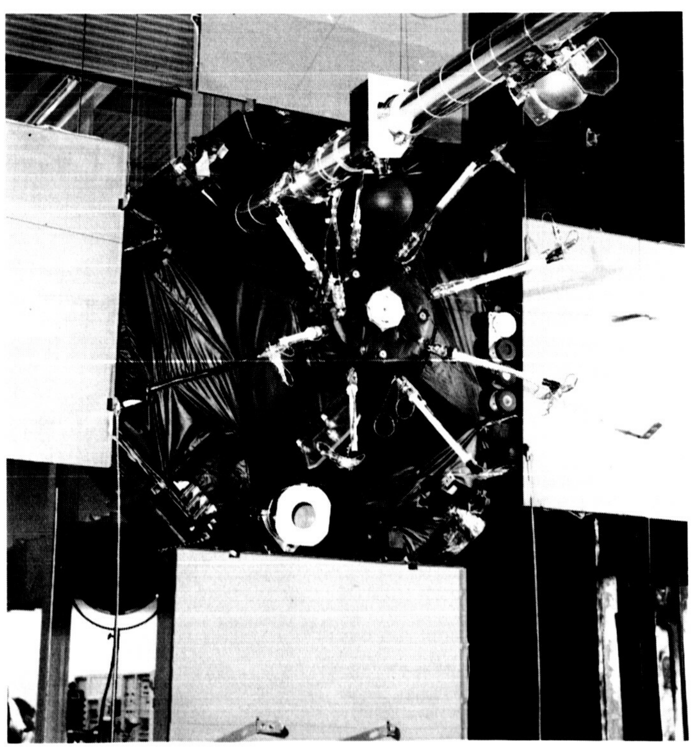

Fig. 5. Upper thermal shield on temperature control model

was coated on one side only with an aluminum layer approximately 3.0 μin. thick. Before the sheets of aluminized Mylar were assembled into blanket form, they were wrinkled by hand-gathering the film into a small bundle and crushing it. The resulting permanent wrinkles or creases fractured the aluminized layer into approximately 160 segments per square inch. The purpose of the crinkling was to minimize the number of point contacts between successive layers. A byproduct of this procedure was the loss of electrical and thermal conductivity across the blanket. After crinkling, the 4- × 6-ft Mylar sheets were laid out flat, with the aluminum side in the same direction, to form 30-layer blankets. The blanket assembly was finished with the addition of the outside layer of black silicone-coated dacron cloth (Fig. 5). The cloth weighed 5.00 ± 0.25 ounces per square yard. Neither the reflectance nor transmittance characteristics exceeded 6% between wavelengths of 0.3 and 0.8 microns. The black cloth was used to render the spacecraft insensitive to spectrum differences between the simulated and the actual Sun. Other advantages of the high-emittance black were low surface temperatures, low reflectance, resistance to ultraviolet radiation, and a durable, easy-to-clean temperature control surface.

Because of material width limits, two such blankets were required for one complete shield. A razor knife was used to cut the blanket to match the paper pattern. All edges of the blanket were then secured by sewing with a sail-type stitch (approximately six stitches per inch) on a commercial textile-type electric sewing machine. With the addition of a nylon zipper, Velcro strips, and lacing flaps, this phase of the operation was complete. The shield was then fitted to a spacecraft for final positioning of the peripheral attachment angles. These 0.016-in.-thick polished aluminum angles, predrilled with 0.060-in.-diameter holes spaced 0.25 in. along the length, were hand-sewn to the blanket edges. This type of attachment retained the blanket thickness for insulating qualities and protected the Mylar edges from solar exposure. The angles in turn attached to upper electronic case screws to secure the blanket assembly to the spacecraft. The final inspection of the shield took place on the spacecraft because its flexibility rendered a dimensional inspection meaningless.

2. ***Retarder shield.*** The retarder shield was mounted under the high-gain antenna and formed the top portion of the upper shield.

The complexity of the spacecraft structure and assembly procedures contributed to the two-part upper shield. When assembled to the spacecraft, the two shields were joined together by a continuous lacing. The retarder shield construction was exactly the same as that of the upper blanket, 30 layers of crinkled ¼-mil aluminized Mylar plus one exterior layer of black dacron. Although small (approximately 24 in. in diameter), this shield contained 18 cutouts for protruding structures and eight flaps for lacing (Fig. 6). This presented the first real compromise between the theoretically poor shield and the practically possible one. The thermal shorts caused by the multitude of holes and the sewing were thermally degrading, but the shield was shaded by the high-gain antenna and was not subjected to direct solar radiation.

3. ***Lower shield.*** The lower shield covered the shaded side of the spacecraft, except for the science platform and necessary protrusions such as connectors, switches, and Sun sensors, in order to prevent an excessive loss of heat to space. Its thermal requirements differed from those of the upper shield because the outer surface would remain at a low constant temperature owing to its space orientation. Heat losses through this shield were expected, but constant and known values had to be assured. Initial indications were that 10 layers of Mylar plus an outside layer of black dacron would be sufficient. The dacron was used on the exterior because of its durability and its possible exposure to solar radiation during midcourse maneuver. The shield was constructed using the same techniques developed on the upper thermal

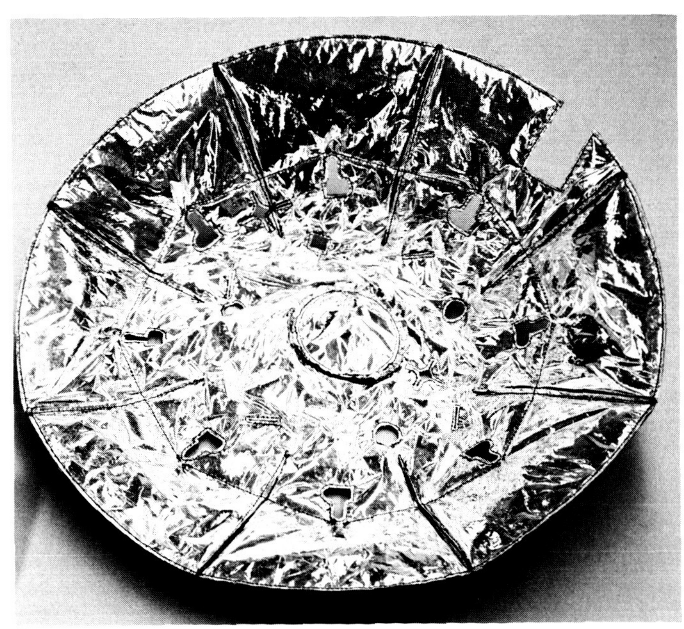

Fig. 6. Retarder shield

shield. The peripheral attachments were again 0.016-in.-thick polished aluminum angles sewn to the blanket and in turn assembled to the spacecraft by lower case screws. In the center of the shield was a large hole for installing it around the planet science platform. The perimeter of this cutout was secured to the spacecraft H-frame structure, allowing the platform to protrude below the shield. Again the final inspection was made with the shield mounted on a spacecraft.

4. Case shields. Active louver assemblies were mounted on five cases, and shielding was necessary in many areas left exposed. The case shields were required to be many sizes and shapes to insulate required areas. Their purpose was much the same as that of the lower shield: to reduce heat losses to a low, known, constant value. The case shields were constructed of multilayer crinkled aluminized Mylar with an outside layer of 5-mil Mylar, which added stiffness to facilitate attachment (Fig. 7). Ten layers of ¼-mil Mylar and one 5-mil sheet with the Mylar side out were sewn together in the same manner as the blankets. To attach the shields, a Velcro hook-and-pile nylon closure appeared to satisfy the requirements. The hook portion of the fastener was bonded to the spacecraft, and the pile strip was sewed to the Mylar shield. The more rigid 5-mil Mylar made it possible to define the case shields dimensionally and measure them mechanically, thus easing fabrication and inspection.

V. Louvers

A. Functional Description

The purpose of the temperature control louvers was to provide an active means of controlling the spacecraft thermal environment by varying the effective emittance of the non-Sun-oriented surfaces of the spacecraft. The requirement for active control was established early in

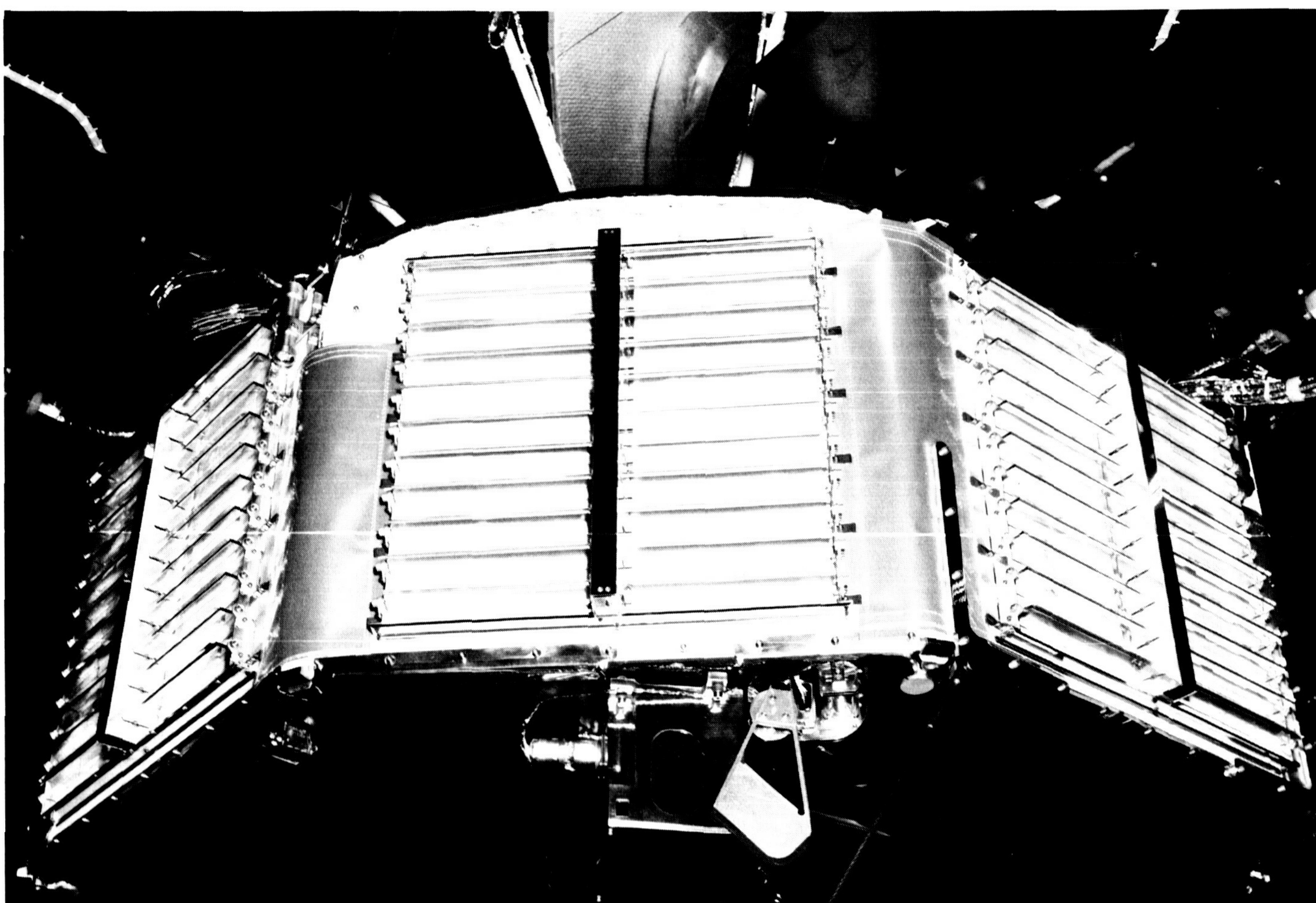

Fig. 7. Multilayer Mylar case shields on temperature control model

the program when it was determined that completely passive techniques could not account for the fluctuations in internal power dissipated and the change in solar intensity. As the spacecraft definition progressed, it became clear that the degree of active control necessary was dependent on the constantly changing internal power profile, and an optimum balance of passive and active control could not be established until late in the design phase. The resultant philosophy was to establish a single louver assembly design and provide an adequate mounting interface with each electronic chassis. The final decision as to which electronic assemblies would require active control could then be made late in the program with a minimum effect on the system.

On the basis of *Mariner II* experience, louvers were selected as the active control technique, with a defined operating temperature of 55°F when fully closed to 85°F when fully open. Acceptable performance limits for the defined operating area were set at 7 w maximum for an average temperature of 50°F and 46 w minimum dissipation at an average temperature of 90°F.

B. Overall Design

As ultimately defined, the *Mariner Mars 1964* spacecraft had six electronic assemblies fitted with temperature control louvers (Fig. 8). The louver area on each assembly was 1.4 ft^2, giving approximately 8.4 ft^2 total

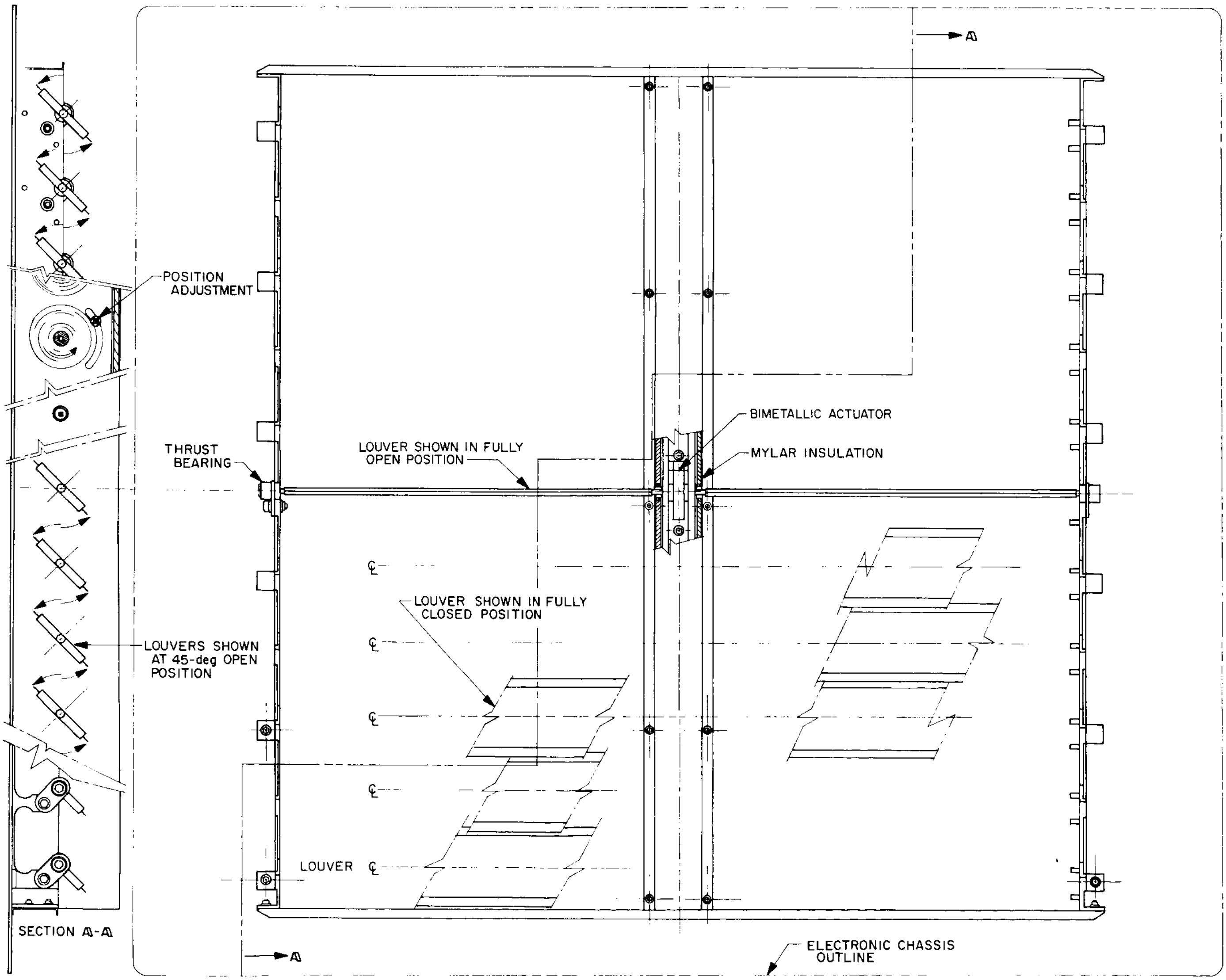

Fig. 8. Louver installation

active control area for the spacecraft. *Mariner II* carried only 1.25 ft^2 of louvers.

A louver assembly consisted of 22 louvers driven in pairs by eleven spirally wound bimetallic elements. Separate actuators, rather than a ganged linkage, were chosen to increase the reliability inherent in independent actuation. These driving elements were situated in a housing in the center of the electronic chassis and were insulated and shielded to provide radiation coupling to the face temperature of the chassis. The bimetallic actuators were sized to provide 90 deg rotation for 30°F temperature change. Although, theoretically, bearing friction in a zero-*g* field should be negligible, consideration for free deflection position error under a reasonably assumed retarding torque forced a tolerance of $\pm$ 5 deg error in position to be placed on the design. The resultant error in equilibrium temperature could easily be ignored.

C. Detailed Design

*1. **Bearings.*** Bearings used as louver pivots were made of a glass-filled Teflon material that was machinable, dimensionally stable, and resistant to changes in friction due to the vacuum environment. The outer bearing housing consisted of the filled-Teflon sleeve running on a hard-anodized aluminum shaft, with adjustable end play being provided by a Teflon TFE thrust pad. The inner bearing support consisted of a filled-Teflon spool (which was also the bimetallic actuator anchor) running on the hard-anodized bore of the actuator housing. The louvers themselves were fitted with square splined nylon stub shafts that mated with the actuator spool, thus allowing louver removal without actuator disassembly. The use of hard-anodized material as a bearing surface against Teflon was a decision based on vacuum data which indicated that unprotected aluminum shafts, when run in conjunction with a Teflon bearing, could possibly suffer an increase in friction as the bearing became impregnated with foreign debris. As a further protection against seizure, bearing clearance was allowed to be quite sloppy – approximately 0.003 to 0.004 in. in both radial clearance and end play.

*2. **Structure.*** Because of the large number of louvers on each spacecraft and the stringent weight restrictions, considerable effort was expended to develop a lightweight louver. As a result of analysis and tests, the selected configuration consisted of a thin-walled aluminum tube with two 0.005-in.-thick aluminum cover sheets bonded to form a narrow box section. The adhesive used was a high-strength, flexible epoxy – EC 2216B/A.

The resonant frequency of a single louver was found to be approximately 400 cps; however, skin thickness could not be reduced below 0.005 in. because of the high probability of damage due to handling. A considerable weight saving was realized in the actuators by using radiation coupling in place of the heavier conduction coupling employed in the *Mariner II* design. As a result of this change in philosophy and other weight optimization, the *Mariner Mars 1964* louver assembly weighed approximately 1.02 lb/ft^2 of active area as compared with the 1.73 lb/ft^2 for the *Mariner II* design.

*3. **Position transducer.*** As a crude indication of proper louver operation, angular position telemetry was provided on the center louver actuator in the form of a semiconductor strain gage. To be compatible with the telemetry channel input, a high-gain, 500-ohm gage was bonded to the bimetallic element and calibrated to give approximately 80 ohms variation when the latter was flexed the equivalent of 90 angular degrees at the shaft. Since these semiconductor gages are extremely temperature-sensitive, it was necessary to calibrate each assembly in several failure-mode louver positions and correlate the resultant reading with the chassis temperature transducer. The estimated accuracy of the engineering measurement was therefore approximately $\pm$15 angular degrees.

D. Assembly Installation

Because of the fragile nature of the assembly, numerous attaching screws were used to align the louver frame to the electronic chassis. Louver blades were then installed individually when a test or flight buildup was scheduled. This procedure was later altered to incorporate a ground handling protective cover to be installed on the completed chassis buildup after it was concluded that damage was likely if the louvers were repeatedly handled. Figure 9 shows the louvers installed on the spacecraft.

E. Test Experience

Following the initial development tests to prove the adequacy of the bearing and louver blade design, 41 louver assemblies were fabricated and subjected to the appropriate type approval (TA) and flight acceptance (FA) tests as outlined in JPL Specifications 31241 and 31242. No structural failures resulted from any of these tests. Functionally, louver performance was within the specified limits, with the measured power dissipation for the TA assembly being 6.9 w at 55°F (fully closed) and 52.6 w at 80°F (fully open).

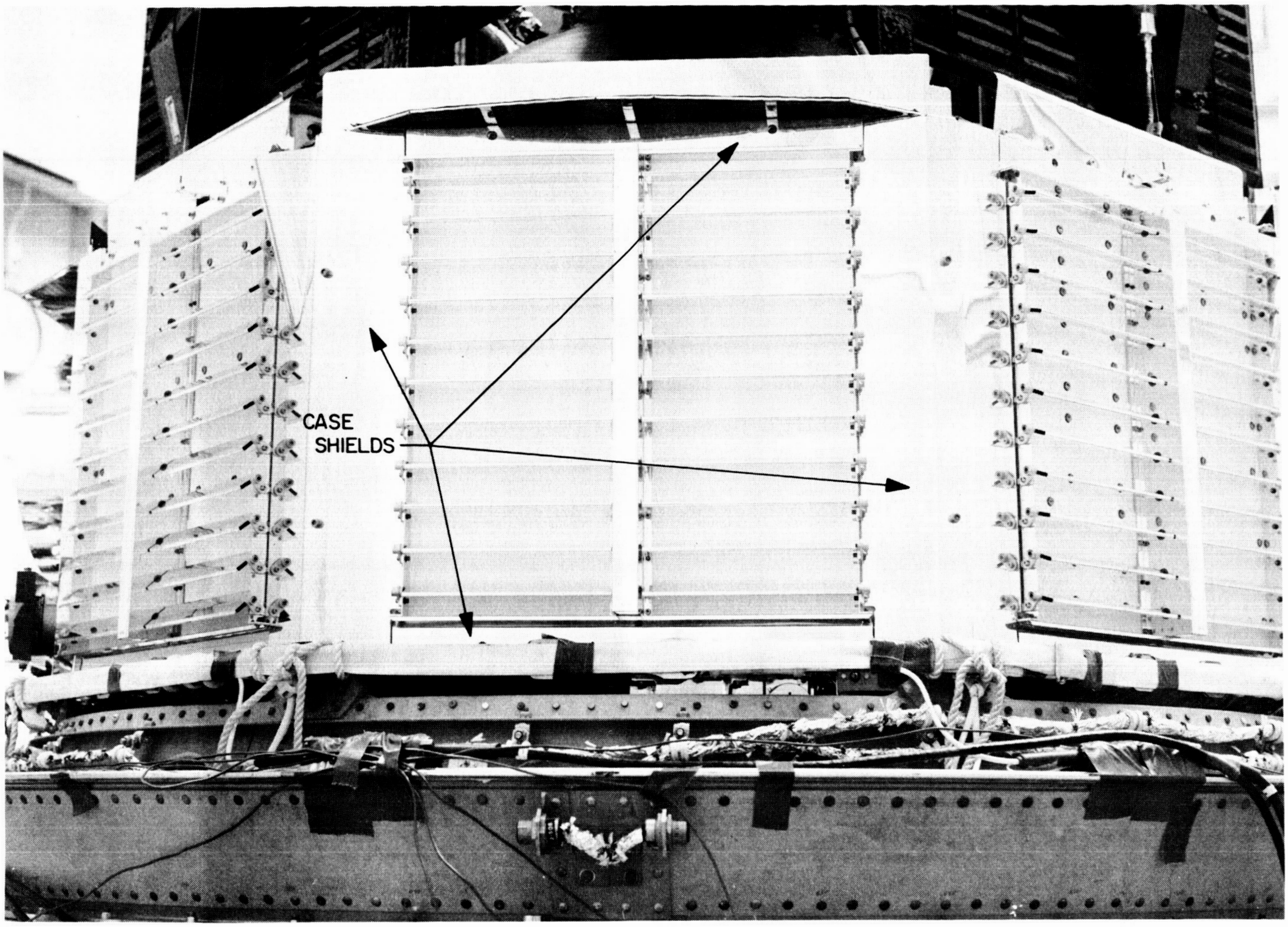

Fig. 9. Louvers and case shields on flight spacecraft

Later Temperature Control Model (TCM) tests indicated not only that more than the five originally anticipated louver assemblies were required, but that the internal power distribution required four of the six assemblies to be recalibrated so that they would be fully closed at 60°F instead of the originally specified 55°F. The 60°F setting was necessary to eliminate thermal gradients in order to obtain operating temperatures closer to the nominal ones. The standard louver assembly configuration and the adjustable range actuator allowed these changes to be made with no design change or retesting required. Later verification during vacuum simulator tests of the flight spacecraft indicated that these adjustments were satisfactory. The flight data on *Mariner IV* confirm proper operation, with the chassis temperature and corresponding louver position telemetry being in direct agreement.

VI. TCM Testing and Resulting Modifications

Based on calculations and prior *Mariner II* experience, the detailed design of the thermal shields, as described in Section IV, was thought to provide an adequate thermal control system. The thoroughness of the test plan, however, was expected to reveal differences between calculated results and actual simulator results. Modification and redesign of shield components were accomplished during the test period as the thermal characteristics of the spacecraft and various shielding approaches became known. The conclusion of the first phase of testing confirmed the adequacy of the basic temperature control design. Subsequent tests, comprising ten separate configurations and 35 test modes, were conducted with the spacecraft positioned in three different Sun attitudes. At the completion of the TCM testing in the JPL 25-ft space simulator, an adequate temperature control shielding hardware design had evolved. Numerous thermal shield modifications were made to attain the desired results. Some of the major shield modifications required during the TCM test and before the Proof Test Model (PTM) testing were as follows:

(1) *Upper thermal blanket.* Performance met design expectations thermally, but the blanket sustained damage to the inside Mylar layer during assembly. A 1-mil sheet of FEP Type A aluminized Teflon was added to increase tear resistance (Fig. 10). Inadequate venting of the air trapped within the blanket was noted. During the chamber pump-down, the blanket ballooned because of internal pressure. Groups of ¼-in.-diameter holes through 10 layers, staggered in successive groups of 10 layers, provided necessary venting.

(2) *Lower thermal shield.* Initially, a shield of 10 layers of Mylar with a black dacron exterior was found to be insufficient. Twenty layers of Mylar were added, then a low-emittance outside layer of aluminum foil, all without significantly changing the shield's efficiency. A 20-layer Mylar shield with a Mylar outside layer was used with successful thermal results. The Mylar, however, disintegrated during the test in which the shield was exposed to solar radiation. A 1-mil Teflon outside layer, which yielded the same thermal results, was added to remedy the situation (Fig. 11). Teflon was also added to the inside for tear resistance. Vent holes were added in the same manner as in the upper blanket.

(3) *Peripheral case shields.* These shields assumed many configurations to accommodate the updating of the spacecraft power profile. Methods of securing them went through several design changes in an attempt to improve their efficiency. The use of metal clips to attach the side blankets was time-consuming and presented undesired thermal conduction paths. The number of Mylar layers and the outside surface emittance were changed several times before the desired results were reached. To ensure repeatable performance, the final design consisted of metal shields made of 0.012-in.-thick polished aluminum (Fig. 9). These shields used existing threaded holes for support and were spaced approximately 0.30 in. from the surface of the electronic chassis by nonmetallic spacers. The additional weight of the metal shields was justified by the mechanical simplicity and thermal repeatability of the design.

(4) *Planet science shield.* During the TCM simulator tests, the temperatures of the instruments on the scan platform were below minimum desired operating limits at Mars. The uncertainty of the conduction and radiation paths required a more positive control. A redesign was necessary to slave

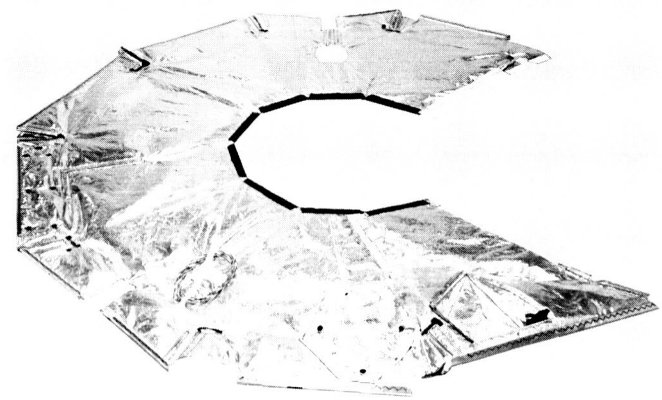

Fig. 10. Upper thermal shield

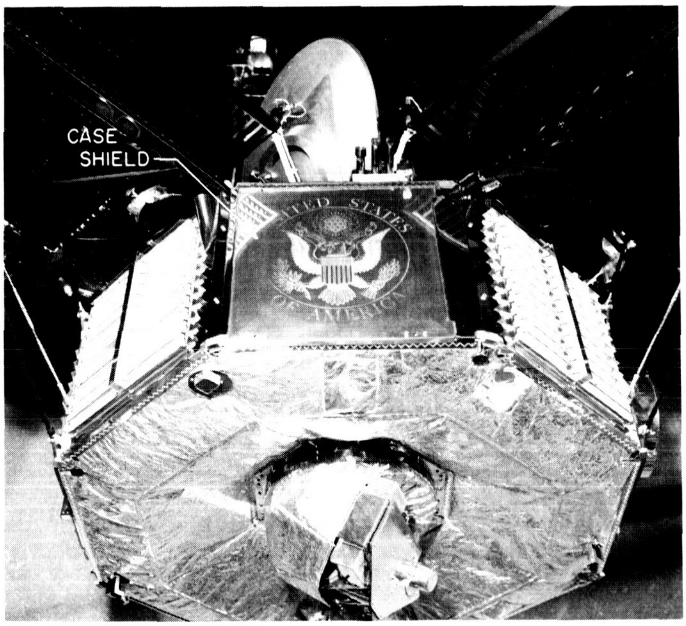

Fig. 11. Lower thermal shield blanket and Case IV thermal shield

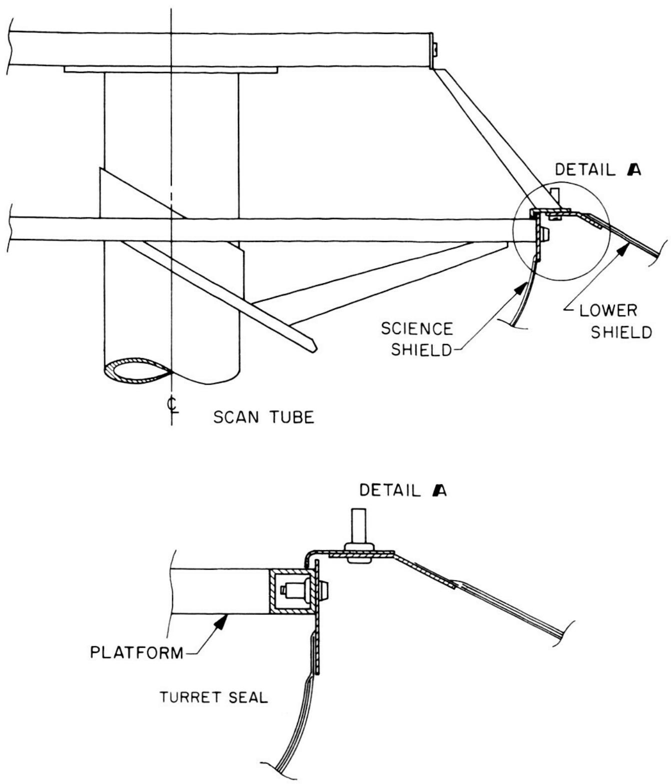

Fig. 12. Science platform turret seal

the platform instruments to the warmer basic octagon structure of the spacecraft. To achieve the radiant coupling desired, a shield was added to enclose the instruments on the scan platform. To allow rotation between the platform and the lower shield, a turret-type design was used. The seal between the turret and the lower shield consisted of two polished aluminum overlapping flanges (Figs. 12 and 13). The turret shield was of the same 20-layer construction as the lower blanket.

To aid the radiant coupling, the instrument housings within the shielded area on the platform were painted with a high-emittance black coating. The exposed portions of the instruments remained a low-emittance polished gold or aluminum (Fig. 14). This method of thermally coupling the scan instruments to the rest of the spacecraft succeeded in keeping the temperatures of the vidicon and sensors within operating limits.

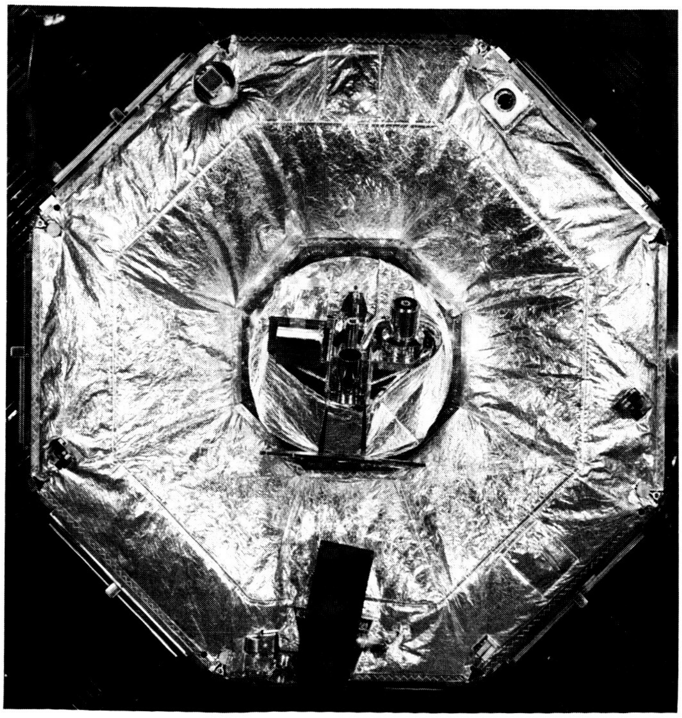

Fig. 13. Lower shield and science platform

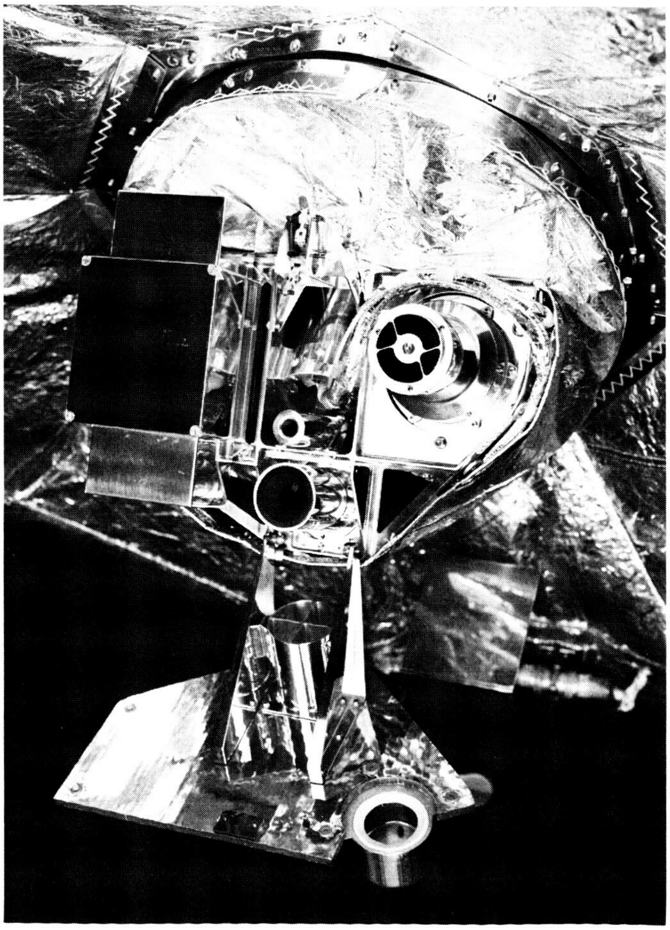

Fig. 14. Science platform turret, with cover open

VII. PTM Testing

All modifications as the result of the TCM simulator tests were incorporated in the Proof Test Model. Before acceptance, all hardware had been structurally qualified on the Structural Test Model. Minor revisions to the case shields were required during the PTM simulator tests. These changes, concerning exposed octagon surfaces, were made by relocating case shields. These substitutions prompted a redesign of the case shields, allowing multiple combinations of basic side-shield components to be used wherever required. Mylar flakes were discovered on the spacecraft throughout the tests, and were traced to the blankets, where the razor knife–cut edges produced tiny slivers of Mylar. Subsequent blankets were heat-cut with a wedge-shaped soldering iron tip at 750–1000°F. At the conclusion of PTM testing, an acceptable flight-type temperature control shielding system had been completed.

VIII. Flight-Spacecraft Simulator Tests and Resulting Special Tests

A. Simulator Tests

The simulator tests were conducted in two phases: systems testing and thermal control testing.

Inspection of thermal shields after the systems portion of the test revealed several problems. The upper thermal blanket had outgassed excessively, the dacron thread used for sewing had shrunk, the exposed Mylar edges had flaked because of ultraviolet exposure, and systems cabling (Fig. 15) had permanently deformed the shield. Any or all of these items could be detrimental to the flight performance.

Before the remaining thermal control tests, a new top blanket was installed, utilizing black-dacron-bound edges to prevent ultraviolet degradation of the Mylar edges. Systems cabling was removed and extreme caution was used in routing the thermal test cabling.

The test was resumed and was completed with successful results. The upper thermal shield did shrink but without adverse effects. Outgassing occurred, and the spacecraft was cleaned without incident. The upper shield was considered flight acceptable because ample time in the simulator assured maximum shrinkage and outgassing. Subsequent tests on flight vehicles used the flight upper shield during the thermal portion only. This avoided the possibility of damage by the additional systems cabling.

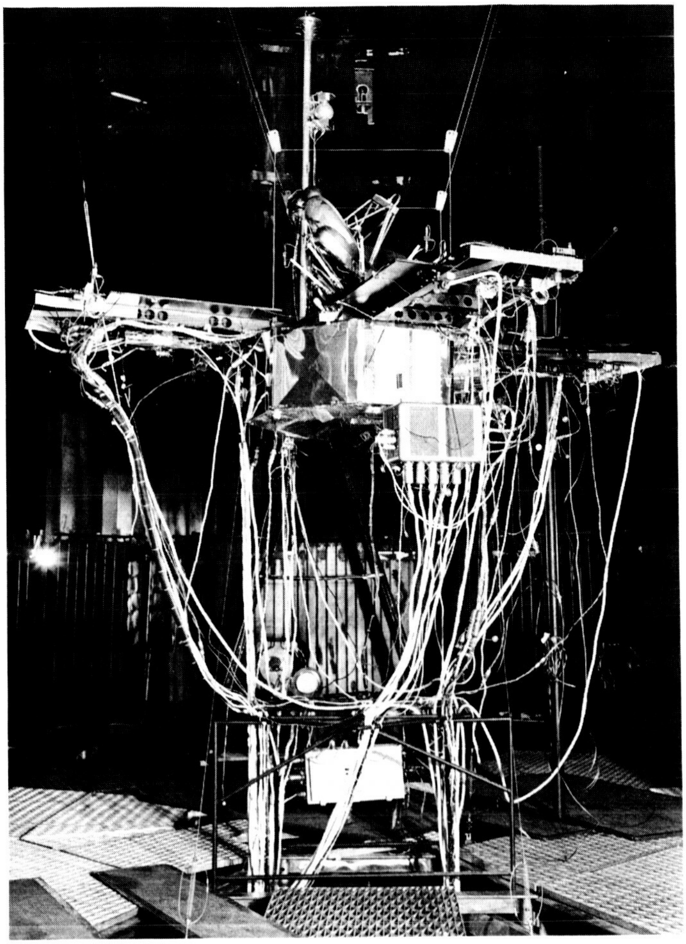

Fig. 15. Systems test configuration of flight spacecraft in JPL space simulator

The first thermal tests also produced modifications because of spacecraft power profile changes. One louver assembly was added and one was relocated to another bay. Case shields were also relocated to accommodate the new configuration. The universal-type design of the case shields permitted this without redesign.

Sun shades were added to the top of the octagon structure to minimize reflected stray light interference with the Canopus tracker. These shields modified the upper thermal blanket and resulted in additional solar inputs being conducted into the bus.

B. Thermal Shield Blanket Conditioning

As a result of the outgassing and shrinkage described above, it was concluded that the spare thermal shield blankets should be subjected to a separate vacuum thermal treatment to outgas the shield materials and stitching, in addition to preshrinking the assembled blankets.

1. Test configuration. A bell-type vacuum chamber 6 ft in diameter and 7 ft high with liquid-nitrogen-cooled walls was used. The development test model (Fig. 16) was used to support the thermal shield. Two banks of 250-w infrared heat lamps were installed above the spacecraft for uniform heating. Temperature data from six thermocouples and from the flight temperature transducer were used to monitor the test. The thermal blanket was weighed and then installed on the structure in a flight-type manner.

2. Blanket test. The chamber pressure was lowered and the upper cooling shroud was cooled to $-70°F$ before the heat lamps were turned on. The upper cooling shroud was intended to trap the outgassing products so they would not contaminate the chamber pumping system. The two banks of lamps were independently controlled by a variable rheostat. After the temperature and pressure stabilized at 300°F and 3×10^{-5} torr respectively, the 24-hour test began. After the blanket was removed from the chamber, it was reweighed and inspected.

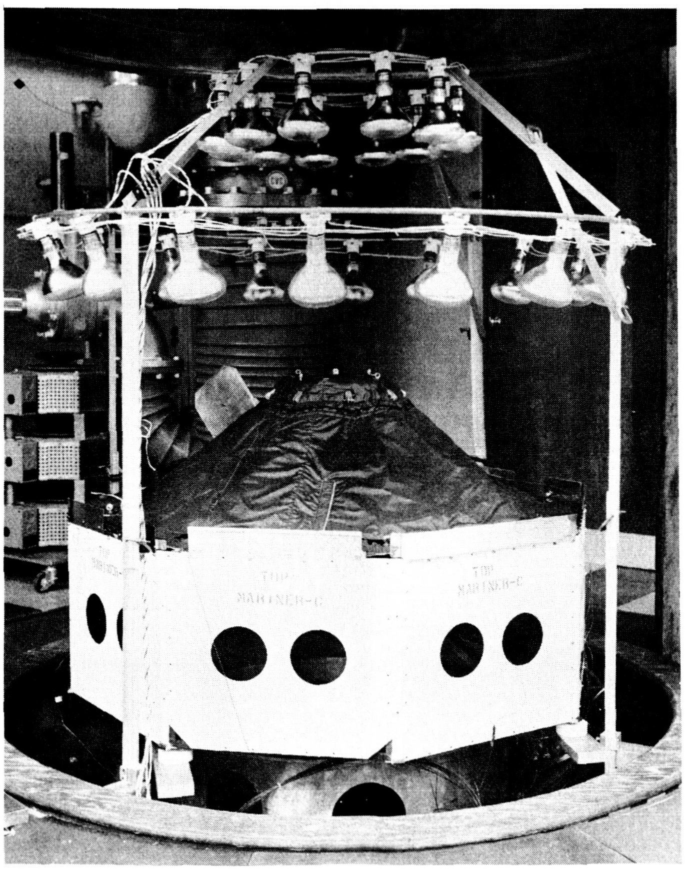

Fig. 16. Spare thermal shield conditioning

3. Results. The thread shrinkage occurred without any apparent damage to the shield blanket. Oil condensation on the chamber walls and reduced blanket weight supported the success of the outgassing treatment.

C. Thermal Shield Blanket Ballooning Test

1. Background. During the shield outgassing and shrinkage tests, it was noted that the thermal blanket ballooned when exposed to rapid decompression, such as it would encounter during the boost phase of the flight. This phenomenon was noted earlier in TCM testing, and vent holes had been incorporated in the blankets. A series of tests was conducted on the upper and lower blankets to verify that the ballooning would not result in structural failure of the blanket or interfere with the functions of other spacecraft components.

2. Test procedures. The upper thermal shield blanket was installed in a flight configuration on the bus structure of the developmental test model (Fig. 17). The bus structure was placed in the 6-ft chamber, and the chamber was pumped down at rates comparable to the launch pressure decay. At the completion of the first test, the structure was turned over and the operation was repeated using the lower shield. Each test lasted for approximately 3 min. During this time, the chamber pressure was reduced at the rate shown in Fig. 18. Photographs were taken through a chamber porthole to record the blanket behavior (Fig. 19).

3. Test results and conclusions. The shields were not damaged and, although they ballooned during the test, they did not interfere with other spacecraft components. Therefore, it was concluded that the vent holes were adequate to prevent structural damage to the shields and that the shield ballooning would not interfere with spacecraft operation after the boost phase of the flight.

IX. Discussion and Recommendations

A. Surface Finishes and Materials

The following recommendations are offered on surface finishes and materials:

(1) Effective shipping containers were provided for completed assemblies, but adequate packaging was not available for some subassemblies and

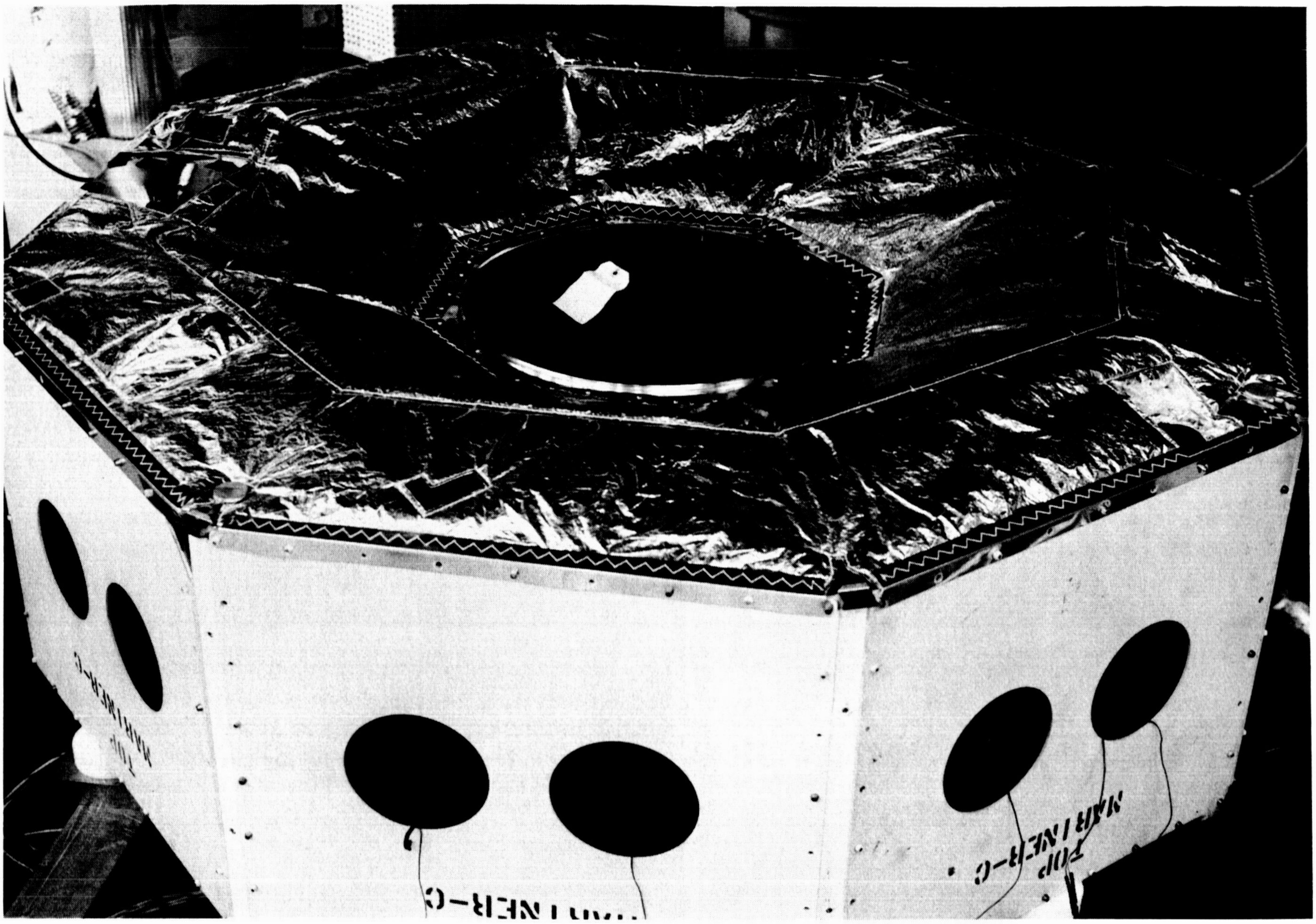

Fig. 17. Setup for lower shield ballooning test

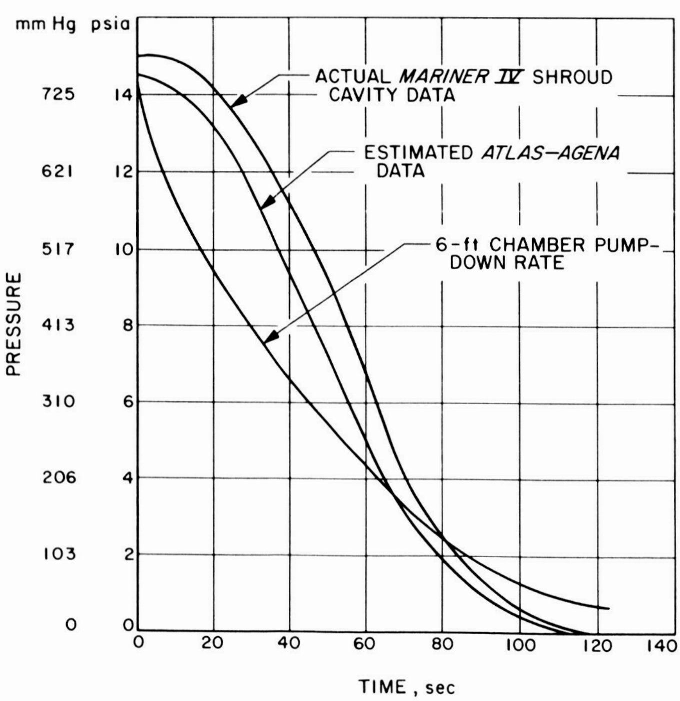

Fig. 18. Pressure decay curves

Fig. 19. Ballooned lower shield at ~70 mm Hg

components, or was not always used. While durability is a consideration in temperature control finish selection, a reasonable degree of handling and transport protection is also necessary. More universal use of suitable containers for in-process hardware should be adopted, providing for protection of thermal finishes as well as for structural and functional performance.

(2) Several incidences of poor adhesion of the coating to the substrate were observed during the program. This defect was usually traceable to inadequate surface cleaning prior to paint application. Since the normal surface preparation techniques, such as gritblasting or chemical etching, are not permissible, a reliable means of detecting surface contamination would be highly desirable for future programs.

(3) A coating lot control system including detailed process and inspection records should be maintained for problem analysis and hardware traceability.

(4) The Permascope used to determine the paint thickness proved highly useful. This device provided a means for assuring that the minimum thickness for optimum radiative properties was achieved without an excess amount being applied. Similar information on coatings should be mandatory on any future programs.

(5) A problem was encountered in maintaining uniformity and reproducibility of the Dow 7 coating for magnesium. In order to assure satisfactory properties for thermal performance, it was necessary to restrict the application of the coating to a single vendor. In addition to the problem with

radiative properties, the Dow 7 did not provide as much corrosion protection as would be desirable. For these reasons, it is recommended that for future programs the desirability of using anodic protection coatings for magnesium be more thoroughly investigated. The thickness and therefore the emittance can be more accurately controlled with this type of coating by maintaining the proper combination of temperature, time, and application voltage.

(6) There is still need for a white coating that (1) is stable to space ultraviolet, (2) has satisfactory engineering properties for Earth-environment handling and, (3) is reliably adhesive. While the inorganic paint mentioned earlier in this report had adequate ultraviolet stability, its use was restricted to those areas where the requirements for optical properties and ultraviolet stability justified the precautions necessary to assure adequate adherence and maintenance of cleanliness.

(7) Some work is being done in the aerospace industry (particularly by the Lockheed Missiles and Space Company) with tape-type temperature control surfaces. This approach has several advantages to offer, including more thorough and reliable quality control, simple and rapid repair procedures, and the capability of using a wider variety of material systems within the constraints of coating cure temperatures. The major disadvantage of this approach lies in the question of the reliability of the surface adhesion. Separation due to expansion of entrapped gases after launch may cause a greenhouse effect. The problem of the application of tapes to the compound curvatures of spacecraft presents another real limitation.

B. Thermal Shields

Provisions should be made for temperature control shielding in the initial spacecraft design. However, as power profile definition early in design is difficult, and passive-type shielding depends on that information, design flexibility is mandatory in any shield design. The *Mariner* temperature control system balanced the active and passive methods by using the versatility of the shielding to comply with temperature requirements as spacecraft testing progressed.

Although all the materials used in thermal blanket fabrication were individually tested for adaption to space environment before use, the finished assemblies should also be subjected to space and launch environment tests such as outgassing, launch pressure decay, and ultraviolet radiation.

The tailored approach to blanket fabrication achieved desired results only after an accurate spacecraft was available for patterns. This method successfully defined the shielding requirements in terms of handling and final button-up procedures. Because temperature control shielding is generally fragile in construction and surface qualities, the handling; installation, and cleaning should be performed by trained technicians.

One problem inherent in the fabrication of multilayer Mylar blankets involves the flakes and slivers caused by cutting during fabrication. These minute particles are potentially capable of interfering with light-sensitive sensors on the spacecraft. Binding all the edges of the thermal blankets contained the particles that may have been present and prevented solar-energy degradation of the otherwise exposed Mylar edges.

Contributing to the successful utilization of multilayer blankets were various design features:

(1) Most of the edges were uniformly secured to retain them during the launch phase. Besides the acceleration forces experienced during launch, the pressure decay tends to expand the blankets because of air trapped within them. Adequate venting and support were necessary for structural survival in that environment.

(2) To avoid miscellaneous random thermal shorts, the blankets were supported above the substructure and components, with a known, repeatable loss at the supporting edges. The intent was to support the shield blankets by the two unwrinkled outer layers, thus preventing local thermal shorts caused by contact with protruding objects.

(3) Blanket shielding design enabled coverage of a large area (approximately 40 ft^2) with little weight penalty (approximately 0.1 lb/ft^2). The light weight, the flexibility, and the low load transmittance characteristics resulted in little or no effect on the spacecraft vibration response modes.

Although aluminized Mylar and Teflon were used exclusively and successfully on the *Mariner Mars 1964* spacecraft for thermal blanket fabrication, several new materials should be considered for future applications.

Polyamide film (H film) with its wide temperature range (-450 to $+1100°F$) and high tensile strength (25,000 psi) is one such promising material.

A multilayer insulation material capable of withstanding elevated temperatures would be desirable, since it would permit utilization of a low-emittance exterior surface. Such a low-emittance surface would minimize the effects of unavoidable heat leaks. It was necessary to use an intermediate-emittance exterior for *Mariner Mars 1964*, since excessive surface temperatures would have resulted from solar illumination during periods of non-Sun-orientation of the spacecraft.

C. Louvers

In addition to confirming the effectiveness of active temperature control by means of louvers, the design incorporated a number of details that should be considered as worthwhile contributions to future design efforts:

(1) A standard-size louver assembly with a simple spacecraft attachment interface should be considered as a convenient method for providing flexibility in the number and location of assemblies required for active control.

(2) Louver blades should be capable of being individually replaced without major disassembly of the unit. This feature is convenient, since there is a reasonable probability of damage to the louvers because of their lightweight, fragile construction and their exposed location on the spacecraft.

(3) Individual louver actuators, rather than ganged linkage, offer the highest system reliability, since a single louver failure has little effect on the performance of the assembly. An additional benefit that ganged operation cannot provide is the ability of individual actuators to compensate for thermal gradients.

(4) Louver actuators should be capable of range adjustment to allow change in the active control temperature during testing.

(5) Spiral wound, bimetallic spring actuators, although recognized as low torque devices, can be effectively used with loose tolerance sleeve bearings to provide low hysteresis positioning.

(6) Although the *Mariner IV* louvers operated as designed, future programs could benefit in some areas of weight saving. As previously mentioned, the louver blades were subjected to extensive weight-saving analysis and are now considered to be as light as can be reasonably handled. To further improve the area/weight ratio, increased louver blade length and width, with the same number of actuators, would be a convenient improvement if space allows.

X. Conclusions

Mariner IV has successfully completed its mission and a large number of engineering temperature measurements are now available to aid in analyzing thermal performance. Flight temperature data, coupled with the known change in solar constant during the mission, have been correlated with ground test data and indicate that the temperature control system has been completely successful.

A total of 36 sensors continuously monitored the spacecraft temperature distribution during flight. Although only two of these were recording actual shield temperatures, conclusions arrived at through spacecraft performance were:

(1) All temperature control shielding survived the launch mode without adverse effects.

(2) Spacecraft temperatures were within ranges predicted from previous space-simulator tests.

(3) The multilayer blankets performed their task as heat barriers.

(4) Polymeric materials, as used, did indeed adapt to the space environment and return expected results.

(5) Polished-aluminum low-emittance surfaces were not thermally degraded by the space environment.

(6) The *Mariner Mars 1964* temperature control system repeated its earthbound test results in space to confirm both the system approach and the materials used.

With a history of successful operation during earthbound testing and the supporting data of flight performance, it is concluded that the integrated active and passive temperature control system has satisfied its functional goals.

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